Technical Report No. 32-659

The Electrically Propelled Lunar Logistic Vehicle

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ABSTRACT

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A parametric analysis of an electrically propelled lunar logistic system is presented. A Performance Index, which is the ratio of the gross payload delivered to lunar orbit during the lifetime of the nuclear-electric powerplant to the total weight placed into the initial Earth orbit during the same period of time, is evaluated. Curves describing the Performance Index as a function of the powerplant mass fraction (and the corresponding thrust device specific impulse) for various round trip flight times and powerplant lifetimes are presented. For any given case, there is an optimum allocation of mass to the powerplant and these optima are presented in summary curves showing the best Performance Index obtainable with a given powerplant specific weight for a specified flight time and powerplant lifetime. For first generation powerplants, which may be heavier and have a shorter lifetime than ultimately expected, one-way trips may be desirable and applicable results are presented. A detailed description of an electrically propelled lunar logistic system based upon the Saturn V launch vehicle is given and the performance is found to be significantly better than that obtained with chemical or direct-heated nuclear rocket systems using the same launch vehicle.

I. INTRODUCTION

By the end of this decade, manned exploration of the Moon is expected to begin with the Apollo project. This mission concept will allow two men to spend from a few hours to perhaps several days on the lunar surface. After some preliminary exploration of this limited type, a requirement may develop for a lunar base that could support a dozen or so men from six months to a year.

Such a lunar base would present very severe logistic requirements. Heavy equipment and materials for construction, scientific experimentation, and land locomotion would be required in addition to a heavy volume of expendable supplies such as food, water, and oxygen. Thus, the practicality of such a base would depend upon the ability to transport enormous tonnages (by space standards) to the Moon at a reasonable cost.

The needs for most of the supplies probably can be anticipated far in advance. Thus, as long as the freight arrives on a regular schedule, the transit time from Earth to Moon is of secondary importance. This time factor, along with the need for large quantities of supplies, suggests an application for a nuclear-electric vehicle. In addition, it may be desirable to reuse the transfer vehicle since the nuclear-electric power supply and its spacecraft components are costly and represent a heavy weight to launch into Earth orbit, and the powerplant would be expected to have a useful lifetime in excess of a reasonable transit time to the Moon.

It should be added that even though a low thrust vehicle looks very attractive for this mission, a high thrust lunar logistic system with its characteristic flight time of 3 or 4 days will probably be needed for initial construction and emergency purposes.

The purpose of this Report is to parametrically examine the performance of an electrically propelled lunar logistic vehicle and to determine its design characteristics. For this initial look at the vehicle, the goal is to determine the optimum powerplant weight (or power) and thrust device specific impulse and to compare the expected performance with that of competitive systems such as the chemical or direct-heated nuclear rocket types of logistic systems.

II. MISSION ANALYSIS

A. Flight Profile

Since thrusts available from nuclear-electric spacecraft result in accelerations much less than the acceleration of gravity, the spacecraft must be placed into Earth orbit by a high thrust booster. Generally, the altitude of the initial orbit should be as low as possible consistent with safety requirements. Safety criteria for such spacecraft have not been defined but initial orbit altitudes from 200–700 nm have been assumed in various studies. The major effect of this choice is to determine the payload capability of the launch vehicle. Of secondary importance is the change in the propulsion requirements for the low thrust phase of the flight.

An initial altitude of 300 nm is assumed in this study. While admittedly on the low side, the performance of the large Saturn launch vehicles is reasonable for this altitude. At present, the second stages of the Saturn vehicles are not restartable and, hence, cannot provide any payload for altitudes near the upper end of the above range. Thus, a requirement for higher initial orbits may require booster modifications or the addition of a "kick" stage.

The spacecraft is assumed to be composed of two major components, the *bus* containing the power supply plus the spacecraft systems; and the *payload module* containing the lunar surface payload, the deorbiting rocket and propellant, and the propellant and tankage for the electric thrust devices. These two major components will generally be launched on separate boosters —perhaps even of different types. After rendezvous and assembly the entire spacecraft will be ready for the outbound leg of its flight.

The application of low thrust in approximately the same direction as the vehicle's motion will cause it to spiral away from Earth. Just before reaching escape velocity, the thrust will be cut off, and the vehicle will coast for about 5 days to the vicinity of the Moon where the thrust will again be applied to achieve capture by the Moon and to spiral into a low orbit of 50-nm altitude. At this time the *lunar orbit payload* consisting of the *surface payload,* the deorbiting rocket, and its propellants and controls will be detached and the rocket used to land the surface payload.

The electrically propelled vehicle then will go through a similar sequence of events and return to its original 300-nm Earth orbit and rendezvous with another payload module launched from Earth. The entire sequence can then be repeated until the lifetime limit of the nuclear-electric powerplant is reached.

For applications such as returning men or scientific samples to Earth there may be a requirement for carrying payload on the return leg of the flight. This mode has not been studied because it adds another parameter to the analysis and because there is little consensus on what the return requirements will be. An important result of return payload requirements, however, is that it makes the round trip look more attractive relative to the one-way mode.

B. Weight Breakdown

In order to get to the heart of the problem without overly complicating the parametric analysis, a relatively simple weight breakdown has been selected. The weight of the vehicle, when assembled and ready to leave its initial Earth orbit, is assumed to be made up of the spacecraft bus W_B and the payload module W_M. Thus,

$$W_1 = W_B + W_M \tag{1}$$

The weight is broken down in this way since only the payload module has to be launched for trips other than the first.

The bus weight, in turn, is broken down into

$$W_B = W_{pp} + W_{st} \tag{2}$$

where W_{pp} is the powerplant weight and W_{st} is the *structural* weight of the bus.

The powerplant weight is defined to include all the necessary power conditioning equipment for the thrusters, and this weight is related to power by the specific weight, α. Thus,

$$\alpha = \frac{W_{pp}}{P} \tag{3}$$

where P is the power available for the *thrust devices.* Inefficiencies in the power conditioning equipment, other electrical loads, and power conditioning weight will make the specific weight as defined by Eq. (3) somewhat higher than is sometimes quoted in the literature. The specific weight is treated parametrically in the analysis which follows, whereas the *powerplant fraction,* which is defined as the ratio of the powerplant weight to the initial weight W_1, is one of the variables to be determined.

The structural weight introduced in Eq. (2) includes the guidance, communications, and attitude control systems; the electric thrust devices; and the actual structure. The weights of these various components depend, in a complex way, on a great many variables and to include them would greatly complicate this analysis. Fortunately, the total weight of these systems should not exceed 10% of the initial weight of the transportation system under consideration and, hence, all of them can be lumped together as *structure*. The total weight will be characterized by a structural weight fraction such that

$$W_{st} = f_{st} W_1 \tag{4}$$

and this fraction will be treated parametrically in the analysis. It should be noted that the weight of some of the electronic systems and the other systems included under *structure* is relatively low and that the performance would not be changed significantly even if their weights were included in the payload module weight, below, instead of the bus weight. In this way they could be replaced on each trip—which would result in a higher reliability at the expense of higher procurement cost. This replacement feature might be particularly attractive for the first generation vehicles.

Returning to Eq. (1), the payload module weight can be broken down into

$$W_M = W_{pl} + W_p + W_t \tag{5}$$

where W_{pl} is the payload weight which is deposited in *lunar orbit*, W_p is the propellant weight, and W_t is the propellant tankage weight.

The payload weight and the propellant requirements are determined by the low thrust propulsion requirements described in the next section and in the Appendix. The tankage weight is related to the propellant weight by

$$W_t = kW_p \tag{6}$$

As will be seen, only ion engines are of interest for this mission; so, for cesium propellant, $k = 0.06$ is assumed. Although estimates range from 4 to 10%, the above value is representative of the propellant-weight-dependent component of the feed system. Furthermore, this effect is relatively minor, so it was not felt to be worthwhile to treat it parametrically.

C. Method of Calculation

The round trip flight profile, as discussed above, can be divided into 4 phases of powered flight and 2 coast periods. Figure 1 indicates a numbering system useful for denoting the spacecraft condition at the end of each of these periods. This figure is not drawn to scale and for reasons of clarity the large number of nearly circular revolutions of the escape spiral are not shown.

The initial weight, W_1, as defined in Eq. (1), is assumed to be given. (Since at this stage only ratios are important, the initial weight can be assumed to be unity.) The final weight, W_6, is given by

$$W_6 = W_{pp} + W_{st} + W_t \tag{7}$$

since the payload is to be dropped off at the Moon, and one wants to return with essentially no propellant. (A small reserve can be lumped in with the tankage allowance.) The payload weight is dropped off at the Moon and, thus, is given by

$$W_{pl} = W_3 - W_4 \tag{8}$$

The Appendix presents the detailed equations giving the flight time and propellant requirements for the various phases of the flight after all the parameters which define the spacecraft have been specified (i.e., powerplant specific weight; powerplant weight fraction; structural weight fraction; and thrust device specific impulse and efficiency). Basically the method of solution involves the direct calculation of the Earth-to-Moon portion yielding W_3 and the corresponding flight time. Since the payload is not yet known, the return flight is essentially "flown" backwards starting with W_6, which is known or assumed, and ending with a determination of W_4, and the flight time for this leg. Then the payload can be calculated with Eq. (8).

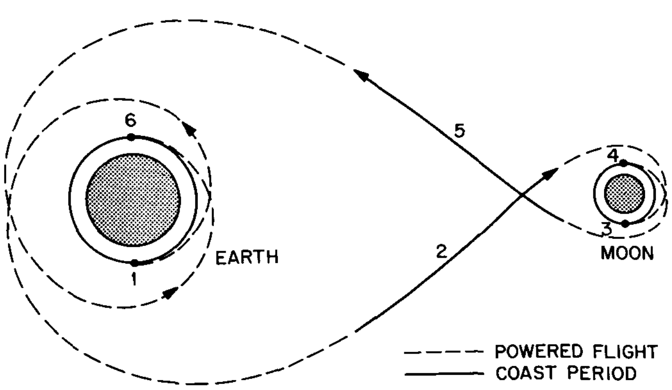

Fig. 1. Schematic representation of phases of flight

It will be noted that W_6 depends on the tankage weight which is not known in advance because the propellant requirements are unknown. Thus, an iterative solution is required to satisfy all the equations. Since the tankage weight fraction is relatively small these iterations converge very rapidly, but the converged solution may not give the desired total flight time. The powerplant fraction is then varied in order to converge on the desired flight time.

The round trip flight times of interest have been chosen as submultiples of the powerplant lifetime which has been assumed to be a multiple of 400 days (just under 10,000 hr). The number of round trips is designated by the variable N; and, as implied above, only integral numbers of round trips are considered. Thus, at the end of the powerplant lifetime the spacecraft has returned to Earth orbit. This generally appears to be unnecessary and if not done would allow additional payload to be carried on the last trip. However, this mode of operation was not considered because:

1. It makes little difference in the performance for more than 2 trips
2. There may be logistic conveniences in having standardized payload modules
3. The powerplant "lifetime" may conveniently be interpreted as a "service life" with the vehicle ending up in Earth orbit for maintenance
4. This simplification allows the reader to determine the entire weight breakdown of a vehicle from the data contained on a single graph of the results

Another option which might be followed in an actual vehicle is to stage the tanks as they are emptied. Again the gain is slight and would unnecessarily complicate the analysis. Also, the practicality of such a scheme depends intimately on the spacecraft configuration and propellant feed system.

D. Thrust Device Performance

The detailed results depend on the assumed performance of the electric thrustors. The performance could be treated parametrically, but this would add another degree of freedom to the analysis. Hence, a variation of efficiency with specific impulse has been assumed as shown in Fig. 2.

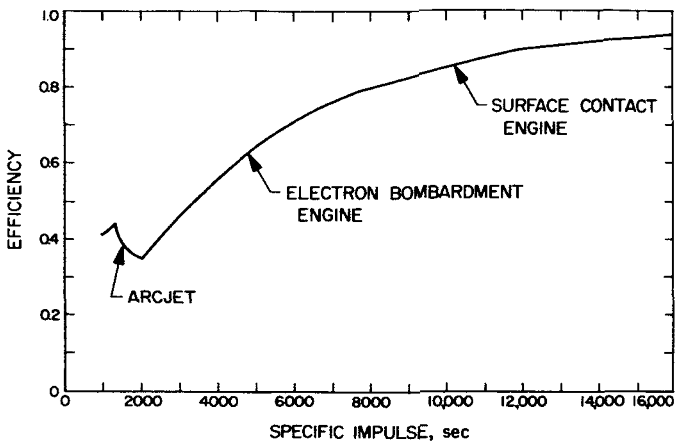

Fig. 2. Thrust device performance

Estimates of thrust device performance vary widely. In this study every attempt has been made to use conservative estimates based on measured or confidently calculated values from *proven* engines. The values used are lower than those usually found in electric propulsion studies.

The values of specific impulse ranging from 2000 to perhaps 8000 sec are based on the cesium propellant, electron bombardment engine under development by R. C. Speiser of Electro-Optical Systems, Inc., (Ref. 1). The values shown from about 4000–7500 sec have been *measured* and the performance down to 2000 sec has been extrapolated as suggested by Mr. Speiser. The high end of the specific impulse range is covered by the cesium surface contact type of engine, and the values shown are generally in agreement with other estimates. Over the range from 6000–9000 sec the performance of the electron bombardment and surface contact engines is comparable, so the actual choice of engine will probably depend more on lifetime and ease of operation considerations than on a couple of percent difference in efficiency.

Some recent arcjet engine data are also shown (Ref. 2). They appear to cross over the electron bombardment data at a specific impulse of about 2000 sec. Since the results which follow show the optimum specific impulse to lie considerably above the arcjet regime, it was not felt necessary to define the cross-over region more accurately.

E. Performance Index Concept

It is necessary to introduce some sort of performance criterion which allows the various spacecraft to be readily compared. Simple criteria such as payload per trip or payload divided by trip time may be of interest for some situations but do not lead to a minimum cost condition. The desired result is to minimize the cost per pound of payload on the lunar surface. Such a criterion is critically dependent on a large number of cost assumptions relating to R and D, learning curves, powerplant costs, launch costs, etc., and no two people will agree on what values should be used. Consequently, it was decided to use a Performance Index (*P.I.*) which is the ratio of the total payload delivered to the Moon divided by the total weight injected into the initial Earth orbit during the entire lifetime of the powerplant. As will be shown later, the payload on the lunar surface can be easily related to the payload in lunar orbit; so, for convenience, the Performance Index is based on the payload in lunar orbit. Thus, in terms of the previously introduced notation,

$$P.I. = \frac{NW_{pl}}{NW_M + W_B} \tag{9}$$

This criterion has the advantage of allowing the reader to recalculate the entire weight breakdown of a vehicle from the data presented on a single graph in the next section. These results can then be used to calculate the Performance Index for powerplant lifetimes other than those presented below, or they can be used to define a new Performance Index based on favored cost figures. Another consideration is that Eq. (9) approximates the cost effectiveness criteria under conditions where the launch costs are predominant—and this is approximately true for the vehicles under consideration.

III. RESULTS

A. Round Trip Missions

Figures 3–6 show the Performance Index as a function of the powerplant fraction (or the corresponding specific impulses) for various round trip flight times for powerplant weights of 20 and 30 lb/kw and structural weights of 5 and 10%, which cover the range of *realistic* estimates. All these curves are based on an assumed 400-day powerplant lifetime. The specific impulse at a few points is indicated, and it varies approximately linearly with powerplant fraction for a given flight time. Even for relatively short flights the optimum specific impulse is seen to be greater than 4000 sec.

These curves all exhibit a rather broad maximum. In this region one is effectively trading powerplant for propellant on an almost one-for-one basis. The curves generally fall off more rapidly for low powerplant fractions than for high. Obviously it is better to be overpowered than underpowered.

The major effect of increasing the structural weight is to decrease the Performance Index by a somewhat greater amount and to increase the optimum powerplant fraction. Little or no change is observed in the optimum specific impulse.

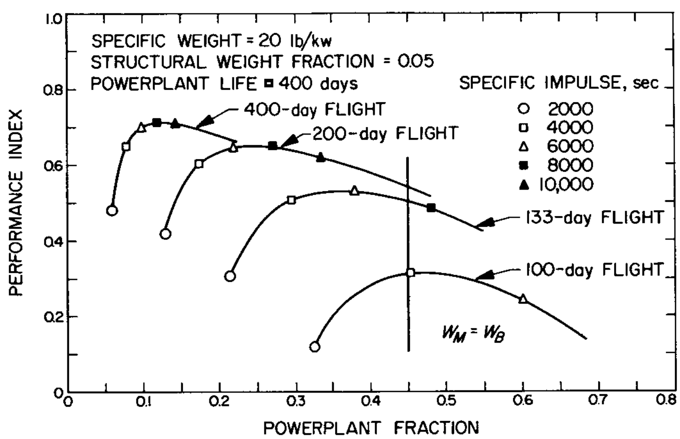

Fig. 3. Round trip performance vs. powerplant fraction; α = 20 lb/kw, f_{st} = 0.05

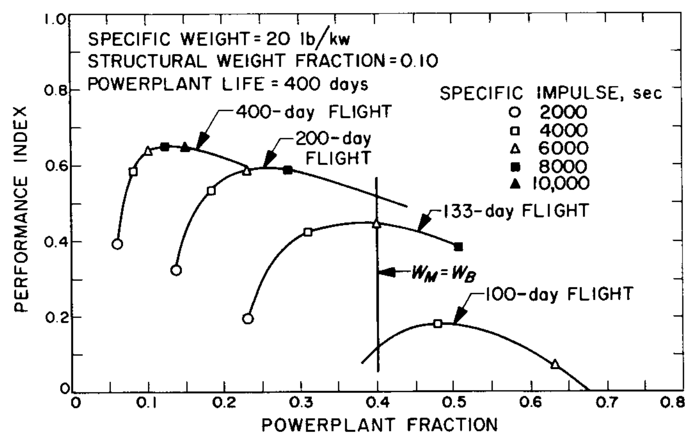

Fig. 5. Round trip performance vs. powerplant fraction; α = 20 lb/kw, f_{st} = 0.10

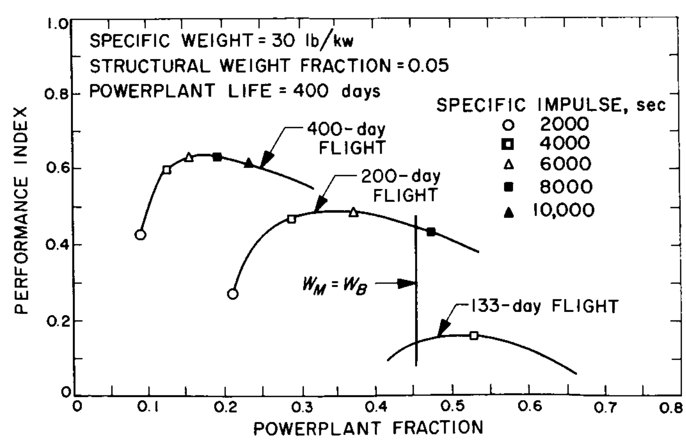

Fig. 4. Round trip performance vs. powerplant fraction; α = 30 lb/kw, f_{st} = 0.05

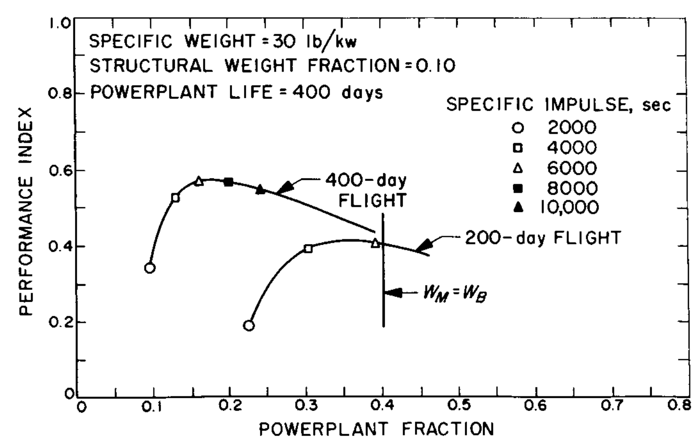

Fig. 6. Round trip performance vs. powerplant fraction; α = 30 lb/kw, f_{st} = 0.10

It can be inferred from these figures that the weight of the payload module and spacecraft bus are of the same order. Since launch vehicles are available only in discrete sizes, it may not be possible to use the maximum performance of the booster for both the payload module and the spacecraft bus launches. One situation where this problem can clearly be avoided is where $W_M = W_B$; and, thus, the *same* type of launch vehicle can be used to its utmost for both types of launches. It is readily shown from Eq. (1) and (2) that this condition is satisfied when

$$f_{pp} = 0.5 - f_{st} \qquad (10)$$

This situation is indicated on these figures. Clearly, if the powerplant fraction is greater than this value the bus weighs more than the payload module and vice versa.

For some logistical planning requirements, it may be desired to know the trip time for the outbound leg of the flight. There is no convenient way to graphically present these data. However, for the range of values of interest (such as in Fig. 3–6), the outbound leg requires between 55 and 80% of the round trip flight time. The higher end of the range is characteristic of those cases having a high payload fraction and vice versa. This rule of thumb should suffice for most purposes. (Some specific examples are given in Table 2.)

The effect of powerplant lifetime is shown in Fig. 7 where some of the previous data have been repeated. It is seen that for longer lifetimes the Performance Index is increased, and the optimum powerplant fraction and specific impulse are higher. This results, of course, because the powerplant is effectively amortized over twice

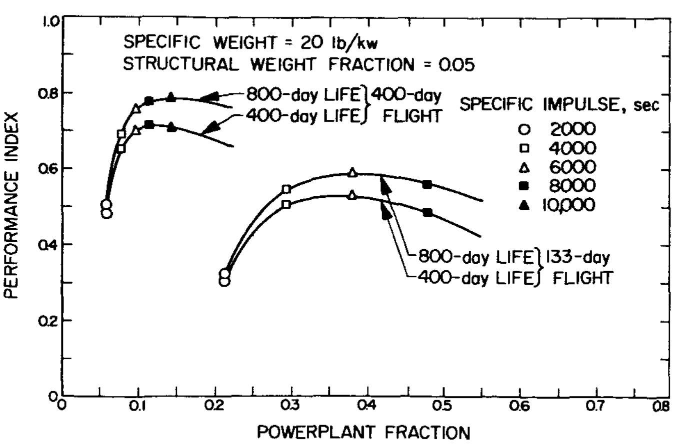

Fig. 7. Round trip performance vs. powerplant fraction; effect of powerplant lifetime

as many trips and can afford to have more weight devoted to it. The differences are smaller for the 200-day trip for a similar reason.

As mentioned earlier, the data are presented in such a way that the entire weight breakdown of the vehicle can be determined from these figures. For completeness, the algebraic equations (which are easily derived from the basic weight breakdown given earlier) will be given. The payload fraction (in lunar orbit) is given by

$$f_{pl} = P.I.\left[1 - \left(1 - \frac{1}{N}\right)(f_{pp} + f_{st})\right] \qquad (11)$$

and after this is calculated, the propellant requirements can be determined from

$$f_p = \frac{1}{(1+k)}\left[1 - (f_{pp} + f_{pl} + f_{st})\right] \qquad (12)$$

The tankage weight is given by Eq. (6). Thus, by using these formulas the entire weight breakdown can be reconstructed and also the Performance Index can be calculated for other powerplant lifetimes. For example, the data presented in Fig. 7 for the 800-day lifetime was generated in this way. The remainder of the results will thus only be presented for one lifetime, namely 400 days, which is likely to be representative of first generation powerplants.

The effect of the powerplant specific weight is best shown by Fig. 8 and 9. Here the Performance Index corresponding to the *maxima* of each of a great many curves like those of Fig. 3–6 are shown as a function of specific weight. These curves clearly show the performance which can be expected for various flight times. Conversely they can be used to determine what powerplant technology (specific weight) is required for the electrically propelled vehicle to meet some standard of performance.

B. One-Way Trips

Under most conditions a single round trip does not appear to be as attractive a flight plan as a one-way trip. The same may be true for two round trips. Therefore, some one-way results have also been included. While results for one-way trips are common in the literature, for purposes of comparison the results are presented here using the same terminology and thrust device efficiency curve.

Figures 10 and 11 show the Performance Index as a function of powerplant fraction for several flight times.

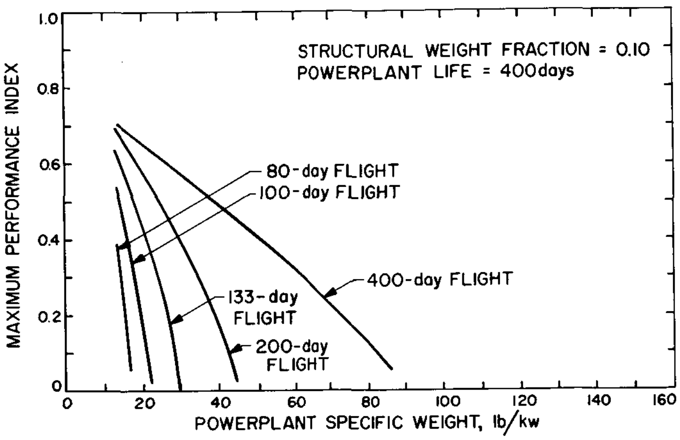

Fig. 8. Maximum round trip performance vs. specific weight, f_{st} = 0.10

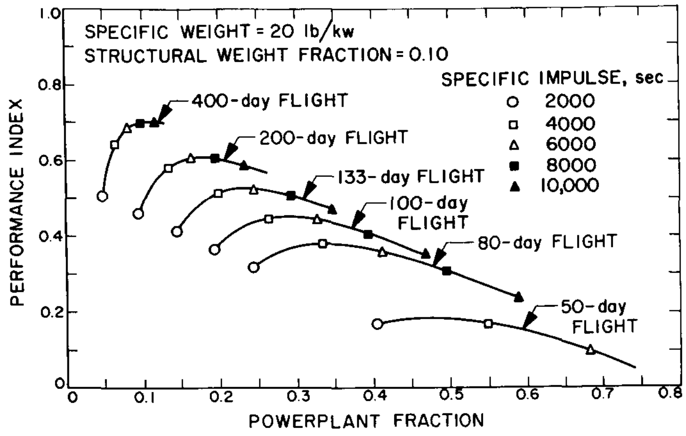

Fig. 10. One-way performance vs. powerplant fraction; α = 20 lb/kw

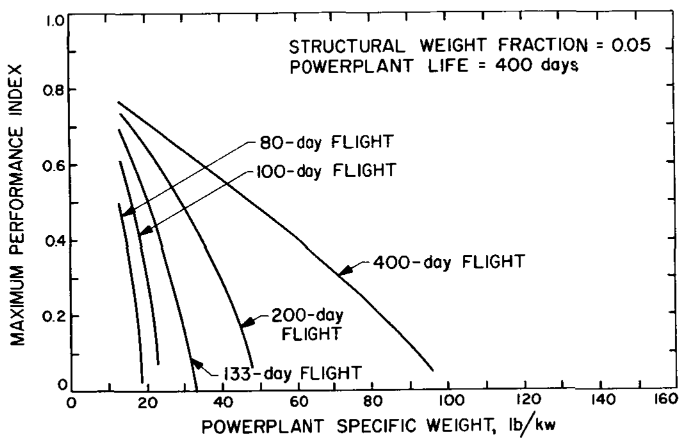

Fig. 9. Maximum round trip performance vs. specific weight, f_{st} = 0.05

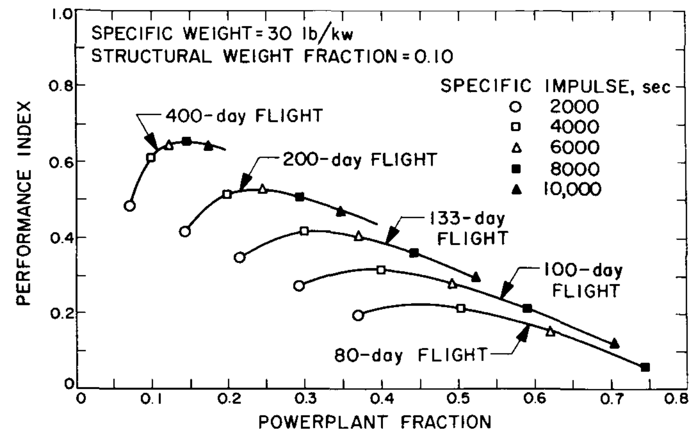

Fig. 11. One-way performance vs. powerplant fraction; α = 30 lb/kw

It should be remembered that these are one-way flight times and are not directly comparable with the round trip flight times presented earlier. In addition, it is meaningful to interpolate between curves to get the performance for other flight times (which is not the case for the round trip missions).

The weight breakdown and the method of calculation presented earlier are greatly simplified for the one-way flights. First, it should be noted that the Performance Index becomes identical with the payload fraction. Second, the *iterative solution for tankage weight is eliminated*; and third, an increase in structural weight results in an exactly corresponding decrease in the Performance Index. For the latter reason, the results are only presented for a structural weight fraction of 10%. Any other value can be obtained immediately by a vertical shift in the curves.

IV. COMPETITIVE SYSTEMS

A. Deorbiting and Landing Requirements

A great deal of study has gone into the analysis of techniques for soft landing from lunar orbit (Ref. 3 and 4). There are generally two modes which are considered

1. A direct descent in which the retro rocket burns continuously from orbit until set-down
2. A Hohmann transfer in which the original circular orbit is converted to an elliptical orbit with a very low periapsis altitude (say, 10 km), and then a retro maneuver slows the spacecraft into a hover and final set-down

Generally, the Hohmann transfer technique results in lower energy requirements, but the differences are not large. The details depend on the thrust level and on the amount of control in the engines.

Generally speaking, a total velocity increment requirement of 2.0 km/sec is adequate to land a payload on the surface from an initial 50-nm orbit. The gains obtained by going to lower orbits with electrical propulsion are minimal, and there is no sense in increasing guidance problems by trying to skim the mountain tops. For the long flight times involved with the electrically propelled vehicles, it is not practical to consider cryogenic propellants. Thus, storable propellants with a specific impulse of 310 sec will be assumed. Since the deorbiting rocket must provide a general purpose payload compartment and must be compatible with the electrically-propelled bus, it may not be possible to use the propellant tanks as an integrated structure. In addition, this deorbiting system must provide its own guidance equipment. Therefore, a fairly conservative structural efficiency (the ratio of propellant to propellant plus "structure") of 0.85 has been assumed. Using the specified velocity increment and propulsion system parameters, it is found that approximately 43% of the weight in lunar orbit is deposited as payload on the lunar surface.

This factor can be used to convert the Performance Index introduced earlier to relate to the payload on the lunar surface. Conversely, the inverse of this deorbiting mass fraction can be used to convert the weight any competing system can place on the surface to the equivalent weight in lunar orbit for the electrically propelled system. This latter approach will be followed.

B. The Chemically Propelled Lunar Logistic Vehicle

The most likely system to be developed is the Lunar Logistic Vehicle (LLV) under consideration by Marshall Space Flight Center and others. It is based on the Saturn V launch vehicle being developed for the Apollo project and consists of the standard 3-stage Saturn V with 2 additional stages—one for the retro maneuver into lunar orbit and the other for the landing maneuver. It is capable of placing a 28,000-lb payload on the lunar surface (Ref. 4).

By using the deorbiting mass ratio discussed previously, it is found that this payload is equivalent to a weight in lunar orbit of about 65,000 lb. Since the basic Saturn V launch vehicle is capable of putting 200,000 lb into a 300-nm orbit, the chemical LLV has an equivalent Performance Index of 0.325. (The chemical LLV never enters 300-nm Earth orbit. However, this is the measure of the launch vehicle performance which is required for defining an equivalent Performance Index for an electrically propelled LLV.)

Referring to Fig. 9, it can be seen that the electrically propelled vehicle can exceed this performance with a 133-day round trip flight for a powerplant specific weight of 20 lb/kw and with a 200-day flight for 30 lb/kw.

C. The Nuclear Rocket Lunar Ferry

One of the most detailed studies of a lunar ferry using direct heated nuclear rockets has been carried out by the Lockheed Missiles and Space Co. (Ref. 5). It is a manned system based on multiple launches of the Saturn V capable of carrying cargo and passengers to the Moon and of returning passengers to Earth. It assumes the use of a *reusable* chemically propelled shuttle to transport the crew and cargo from lunar orbit to the surface. The differences in the operating mode of this vehicle make it somewhat difficult to calculate an equivalent Performance Index for purposes of comparison.

Under their concept, the unfueled ferry rocket and its command module are launched by a single Saturn V booster. Two additional Saturn V launches are used to supply the fuel and cargo to the ferry. In addition, other vehicles are used for maintenance and to shuttle the crew and passengers to Earth orbit. The gross weight of the vehicle as it leaves Earth orbit is 547,000 lb.

After arrival in lunar orbit, the ferry is met by a chemically propelled shuttle vehicle which picks up the cargo modules and passengers and carries them to the surface. The *net* payload on the surface is 28 men and 32,600 lb of cargo for a total of 38,200 lb. Using the previously discussed deorbiting mass ratio, the above figure is equivalent to 90,000 lb in lunar orbit. Neglecting the initial boost to put the ferry in Earth orbit and the Earth shuttle vehicles for the crew and passengers, two Saturn V launches are required for the mission. Since these 2 vehicles can put 400,000 lb in the orbit desired for the electrically propelled spacecraft, the equivalent Performance Index is 0.225.

This method of comparison overly penalizes this nuclear rocket system. It does not properly account for the requirement of supporting 30 men (including 2 crew members) and for the ability to return them to Earth orbit. Perhaps a fairer comparison would be to count the total useful weight in lunar orbit of 132,000 lb (which includes the command module, men, and cargo modules) plus propellant and structure savings of 68,000 lb which would result if the ferry did not have to transport the command module and men back to Earth. Thus, a total lunar orbit weight of 200,000 lb could be achieved which corresponds to an equivalent Performance Index of 0.50. Actually, Lockheed considered 10 round trip flights so that the extra booster required for launching the ferry reduces the Performance Index by 5% to 0.475. This value, however, is optimistic since their study is based on a conceptual design of a second generation metallic core nuclear rocket which has a very low weight and operates at a specific impulse of 830 sec.

Actually the above mode of operation is not particularly attractive. It has been shown (Ref. 6) that for situations where there is no requirement for a return payload (which is probably always the case for unmanned vehicles), there is no advantage in attempting to return the nuclear rocket for reuse. In fact, there is even a slight weight penalty in trying to reuse it.

Referring to Fig. 9, it is seen that this measure of performance is nearly equalled by the electrically propelled vehicle for round trip flight times of 133 and 200 days for powerplant specific weights of 20 and 30 lb/kw, respectively. The performance is greatly exceeded if flight times of 200 and 400 days are chosen for the same specific weights.

V. DETAILS OF SATURN V CLASS VEHICLES

The Saturn V launch vehicle has a cost effectiveness about three times better than the Saturn I-B. This advantage, coupled with the anticipated requirement for very large quantities of supplies, leads immediately to the exclusive consideration of the Saturn V booster. As mentioned earlier, this booster is assumed to have the capability of putting 200,000 lb in the 300-nm initial orbit.

In the previous comparison with competitive systems, four different combinations of powerplant specific weights and flight times were found to be attractive. In addition, a case with a 100-day flight time will also be considered because it exhibits some interesting behavior. The results which follow are based on a 10% structural weight fraction. Round trip performance information can be found in Fig. 5 and 6 and one-way performance in Fig. 10 and 11.

Table 1 lists some interesting results for these cases. Summary data are given for the *optimum* condition as given by the Performance Index criterion. When the powerplant fraction is less than 0.40, the module weight is 200,000 lb, and the bus comes out something less. For the 100-day flight time where the optimum powerplant fraction is greater than 0.40, the bus is chosen to weigh

Table 1. Round trip performance details[a]

Powerplant specific weight, lb/kw	Round trip flight time, days	Condition	Powerplant fraction	Specific impulse, sec	Performance index	Power, kw	Lunar surface payload, lb	Lunar surface payload rate, lb/day	Lunar surface payload per Saturn V launch, lb
20	100	"Optimum"	0.479	4,000	0.179	8,270	15,000	150	12,000
		$W_M = W_B$	0.400	2,800	0.113	8,000	12,100	121	9,700
20	133	"Optimum"	0.376	5,500	0.447	7,170	50,000	375	37,500
		$W_M = W_B$	0.400	6,000	0.446	8,000	51,100	383	38,300
20	200	"Optimum"	0.266	7,500	0.593	4,190	65,700	328	43,800
		$W_M = W_B$	0.400	11,500	0.520	8,000	67,100	335	44,700
30	200	"Optimum"	0.344	5,000	0.410	4,120	49,300	246	32,900
		$W_M = W_B$	0.400	6,200	0.400	5,330	51,600	258	34,400
30	400	"Optimum"	0.176	7,000	0.573	1,620	68,100	170	34,000
		$W_M = W_B$	0.400	16,500	0.425	5,330	73,100	183	36,500

[a] Structural weight fraction 0.10
Powerplant life 400 days

200,000 lb and the payload module is less. For each case the powerplant size, the lunar surface payload, the payload rate (which is proportional to the total payload delivered, since all the vehicles have a common 400-day life); and the payload per Saturn V launch is presented.

The payload rate is maximized by a 133-day flight at 20 lb/kw and a 200-day flight at 30 lb/kw. (The payload rate for a 133-day flight at 30 lb/kw is zero.) The payload rate can be used in conjunction with any specified lunar supply requirement to calculate the necessary number of logistic vehicles which must be in simultaneous operation.

For each case summary data are also given for when the payload module and bus are both chosen to weigh 200,000 lb ($W_M = W_B$). For four of the cases this means an increase of power (and specific impulse) and a slightly *higher* surface payload. This is a situation where the Performance Index does not give the "true" optimum because the effect of discrete launch vehicle sizes was not included. It will be noted, however, that the performance increase is relatively small and that this improvement is only gained at the expense of a much larger powerplant. Thus, even though a minor performance reduction results, it seems better to use the optimum condition and launch a spacecraft bus which does not fully utilize the Saturn V launch vehicle. For the 100-day flight time case, the equal size launches reduce the total payload delivered, and it is actually better to use a payload module of less than the full booster capability. There may be some exceptions to these trends, but generally these results support the rule-of-thumb that it is better to be overpowered than underpowered.

The lunar surface payload per Saturn V launch can be directly compared with the 28,000-lb capacity of the chemically propelled LLV. It is seen that for all but the 100-day flight time this performance is greatly exceeded. The optimistic performance cited earlier for the nuclear rocket vehicle corresponds to 40,800 lb of lunar payload per Saturn V launch and, thus, is only exceeded at 20 lb/kw.

For these first-generation electrically propelled spacecraft with high specific weight and assumed short lifetime, a relatively few round trips are made. It therefore might be desirable to use them (perhaps initially) in a one-way mode. Table 2 summarizes the performance for one-way trips which take the *same* time as the outbound leg of the round trip missions shown in Table 1. To maintain the simplicity of the one-way mission, the spacecraft was assumed to be fully assembled and supplied on the ground. Thus, the initial weight of the entire vehicle is 200,000 lb in order to match the assumed capability of the Saturn V.

By comparing Tables 1 and 2 it is seen that the one-way mission clearly predominates in two situations. The first, for the 100-day round trip, occurs because the short flight time and the resultant low specific impulse and

Table 2. One-way trip performance details[a]

Round trip				One-way trip				
Powerplant specific weight, lb/kw	Flight time, days	Condition	Outward leg flight time, days	Optimum powerplant fraction	Optimum specific impulse, sec	Performance index	Power, kw	Lunar surface payload per Saturn V launch, lb
20	100	"Optimum"	57.1	0.417	3,000	0.245	4,170	21,100
		$W_M = W_B$	57.9	0.411	3,000	0.251	4,110	21,600
20	133	"Optimum"	84.0	0.318	4,000	0.395	3,180	34,000
		$W_M = W_B$	82.9	0.323	4,000	0.391	3,230	33,600
20	200	"Optimum"	140	0.233	6,000	0.536	2,330	46,100
		$W_M = W_B$	129	0.242	5,500	0.516	2,420	44,400
30	200	"Optimum"	129	0.327	4,500	0.403	2,180	34,700
		$W_M = W_B$	125	0.316	4,000	0.395	2,110	34,000
30	400	"Optimum"	303	0.176	7,000	0.610	1,170	52,500
		$W_M = W_B$	261	0.196	6,500	0.582	1,300	50,100

[a] Structural weight fraction 0.10

payload ratio cause the propellant requirements for the return trip to be nearly as great as for the outbound trip. This requirement is enough to offset the advantage of returning the spacecraft bus. The second instance occurs for the 400-day round trip where the powerplant is not reused and, thus, there clearly must be a penalty for returning it to Earth orbit.

For the 133-day round trip flight it is clearly better to reuse the vehicle. The results for the 200-day round trip flights show a small advantage in payload per Saturn V launch for the one-way mode. However, twice as many powerplants and other spacecraft systems are required in order to provide about the same lunar payload. Thus, in a cost analysis, this small weight advantage would be offset by the extra cost of the spacecraft and a comparable cost effectiveness for the one-way and round trip mission modes should result. This result may mean that a logical development program should initially use one-way trips but as experience is gained and the powerplant specific weight decreases and lifetime and confidence increase, a switch should be made to the round trip mode.

VI. CONCLUSIONS

A performance criterion has been introduced and used to indicate the optimum performance of the electrically propelled lunar logistic vehicle over a wide range of parameters. Analytical approximations have been developed which allow the propulsion requirements for the trip to be determined. The results are presented in such a way that the entire weight breakdown can be reconstructed if desired.

The performance has been compared with competitive systems which also are based on the Saturn V launch vehicle. It has been shown that even with a powerplant specific weight as high as 30 lb/kw, the performance of the chemically propelled LLV is significantly exceeded, and that at 20 lb/kw the performance of an optimistic nuclear rocket system is exceeded. In the power range of interest (4 to 8 Mw) a specific weight of 20 lb/kw should be a conservative estimate; and, thus, nuclear rocket performance should in fact be exceeded.

Another significant conclusion is that under those conditions where the electrically propelled vehicle compares favorably with its competitors, the desired specific impulse lies in the range of 5000 to 8000 sec. This is a range in which there has been a great deal of experience with several types of electrostatic thrust devices. Efficiency data have been *measured* and the lifetime limiting effects are understood. There is confidence that long lifetimes can be achieved.

The use of electric propulsion for one-way trips was also examined and found to give comparable performance to the round trip missions for the conservative parameter ranges of initial interest. One may, therefore, decide to avoid the complex rendezvous and assembly requirements of the reusable system and use the one-way mode with initial weight sized for a single Saturn V launch. Here the desirable power levels range from 2 to 3 Mw, and the specific impulses of interest exceed 4000 sec (which can be handled by the electron bombardment thrust device).

There are areas which require further analysis beyond the scope of this preliminary study. Some fairly detailed cost analyses should be carried out with the sizeable R and D costs included. Also detailed work must be done on spacecraft configuration and packaging, and compatibility established for the Saturn V. Likewise the rendezvous, assembly, and checkout operations must be studied in some detail, and if men are required, radiation dosages must be established. It is also necessary, at a fairly early date, to resolve the nuclear safety problem as it applies to the selection of the initial orbital altitude. If the required initial altitude turns out to be high (say 600 or 700 nm), the attractiveness of this system will be greatly reduced. Not only is the launch vehicle performance greatly degraded but also, if men are required for assembly and checkout, the requirements for transporting and shielding them against Van Allen belt radiation are also greatly increased and add to the mission cost. Lastly, but of great importance, is the need to decide upon some specific lunar base, with its associated support requirements, in order to realistically assess the effect of flight time and unreliability on the ferry operations.

APPENDIX

I. TRAJECTORY ANALYSIS

No one has yet been able to compute fully optimized low thrust trajectories for lunar missions; but even if they could be computed, it would not be justifiable to do so for a parametric study such as this. Partially optimized trajectories are available in the work of London (Ref. 7), although his results are presented in tabular form and are not convenient for numerical calculations. It is felt that London's work is sufficiently accurate for this study and analytical equations which accurately approximate his results are developed below.

An object in orbit at the Moon's distance is nearly at escape energy. Thus, one would expect that the low thrust trajectory required for transfer to the vicinity of the Moon would not be very much different from an Earth escape trajectory. The same can be said about the capture and departure phases of flight which take place in the Moon's sphere of influence. Therefore, the method of approach will be to treat the phases of flight as if they were escape or capture trajectories in a single central force field and then to determine a correction factor by comparison with London's results.

It has been shown by Melbourne (Ref. 8) that for tangentially directed constant thrust the time required to escape from a central force field is given by

$$\tau = f(\nu)\,\gamma(a_0)\,\frac{V_0}{a_0} \qquad \text{(A-1)}$$

where V_0 is the velocity of the spacecraft in the low circular orbit and a_0 is the thrust acceleration in this orbit. $\gamma(a_0)$ is a function giving the equivalent free-space velocity increment and for a tangential thrust program is given by (Ref. 9)

$$\gamma(a_0) = 1 - 0.8066\left(\frac{a_0}{g_0}\right)^{1/4} \qquad \text{(A-2)}$$

where g_0 is the local acceleration of gravity (in the low orbit). The function $f(\nu)$ takes into account the mass change of the vehicle, and it is shown to be

$$f(\nu) = \frac{1 - e^{-\nu}}{\nu} \qquad \text{(A-3)}$$

where $\nu = V_0/C$ and $C = I_{sp}g_e$ is the exhaust velocity of the thrust device. Capture maneuvers are equivalent to escape if one considers that the vehicle gains mass as it spirals away rather than losing it. Mathematically the above equations apply to capture if ν is replaced by $-\nu$ in Eq. (A-3).

These equations apply to constant thrust directed along the instantaneous velocity vector of the vehicle (i.e., tangent to the flight path). Constant thrust occurs if the power to the thrust devices and the specific impulse are constant. The power clearly should be constant at its maximum value and a change in specific impulse requires changing the output voltage of the power supply (which may be difficult). Thus, the constant thrust assumption is a good one. The assumption of tangentially directed thrust is justified because it results in values which differ much less than 1% from the optimum value (Ref. 9).

The correction factor is determined by comparing the flight time London presents for each of the four phases of powered flight in the Earth-Moon round trip with the value given by Eq. (A-1). Over the range of specific impulses from 2000 to 10,000 sec and a range of initial accelerations giving round trip flight times from 50 to 500 days, it is found that the mean correction factor is 0.92.

Some scatter is present in the data which probably results from imperfect patching of the trajectories. However, the biggest variations occur in the lunar phases of flight which represent a small portion of the total round trip. It thus appears that the accuracy of the correction is about 1%.

In order to specify the round trip time it is necessary to know the duration of the coast periods. The coast times presented by London do not seem to correlate well with either specific impulse or thrust acceleration. However, for the range of variables of interest, the one-way coast time varies between 4 and 6 days. Therefore, a fixed coast of 5 days was chosen for all trajectories, and the basic accuracy should be about ±1 day.

Using the notation introduced in Fig. 1 and the correction factor presented above, the detailed equations for the 4 phases of flight can be derived. For the Earth escape phase, the equation is

$$\tau_{12} = 0.92 \frac{C}{a_1} \gamma (a_1) (1 - e^{-\nu_e}) \qquad \text{(A-4)}$$

where ν_e is evaluated at the orbital velocity in the initial Earth orbit. The initial thrust acceleration, a_1, is known because the thrust level and initial mass are specified by the (chosen) parameters of the spacecraft. The mass flow rate is given by the thrust divided by the exhaust velocity and is a known constant. Thus, the propellant used can be calculated and the mass and thrust acceleration at point 2 can be determined.

The lunar capture time is formally given by

$$\tau_{23} = 0.92 \frac{C}{a_3} \gamma (a_3) (e^{\nu_m} - 1)$$

where ν_m is based on the final orbital velocity around the Moon. This expression cannot be evaluated because a_3 is unknown. However, a_3 can be expressed in terms of a_2 using the above expression and the expression for the mass flow rate. The resulting equation can then be solved algebraically to give the capture time in terms of a_2. This calculation yields

$$\tau_{23} = 0.92 \frac{C}{a_2} \left[\frac{\gamma (a_3) (e^{\nu_m} - 1)}{1 + 0.92 \gamma (a_3) (e^{\nu_m} - 1)} \right] \qquad \text{(A-5)}$$

It is seen that the function γ must still be evaluated at a_3. However, the function is slowly varying, and the above equation can be satisfied in one or two iterations.

The return trip is handled in a similar manner. However, because the payload dropped off in lunar orbit is as yet unknown, the return trajectory is "flown backwards" beginning with the final Earth orbit conditions, which are known. Thus, the Earth capture time is determined by

$$\tau_{56} = 0.92 \frac{C}{a_6} \gamma (a_6) (e^{\nu_e} - 1) \qquad \text{(A-6)}$$

Knowing the propulsion time, the mass at point 5 can be determined.

Since the mass of the vehicle when it is ready to leave lunar orbit (point 4) is still unknown, the lunar "departure" time must be expressed in terms of a_5. In a manner similar to that used in deriving Eq. (A-5), it is found that

$$\tau_{45} = 0.92 \frac{C}{a_5} \left[\frac{\gamma (a_4) (1 - e^{-\nu_m})}{1 - 0.92 \gamma (a_4) (1 - e^{-\nu_m})} \right] \qquad \text{(A-7)}$$

Again an iterative solution is required because the function γ must be evaluated at a_4.

The above equations allow the propulsion time and propellant requirements for the 4 powered phases of flight to be determined. The payload to be dropped off is the difference in mass between points 3 and 4. This number can be numerically negative and then represents an infeasible choice of spacecraft parameters which physically correspond to a situation in which the ferry cannot even make it back to Earth without picking up some additional propellant in lunar orbit.

II. EQUIVALENT VELOCITY INCREMENT

Besides London's work, the only other low thrust lunar trajectory analysis in the literature is that of Brown and Nicoll (Ref. 10). All the details of their work are not available, but they conclude that the propulsion requirements can be adequately represented by an effective velocity increment which is independent of specific impulse and initial thrust acceleration. It is interesting to compare their result with that of this analysis. Brown and Nicoll consider trajectories which begin in a 300-statute-mile Earth orbit and end in a 20-statute-mile lunar orbit. They conclude that the effective velocity increment is 25,700 ft/sec.

For purposes of comparison, Eq. (A-4) and (A-5) were used to give the propellant requirement for a one-way trip between the same orbits. The effective velocity increment is defined by

$$\Delta V = - C \ln \left(\frac{W_3}{W_1} \right) \qquad \text{(A-8)}$$

The results for a range of specific impulses and initial thrust-to-weight ratios are shown in Fig. A-1. It is seen that there is a variation of about 2% with specific impulse. The range of thrust-to-weight ratios which result in one-way flight times of about 50–200 days is indicated, and the velocity increment varies about 3% over this range. Hence, it is seen that although the assumption of a constant velocity increment is not fully valid, the error is not large. It also appears that the Brown and Nicoll estimate is slightly less optimum in the range of interest than the present (London) results.

It should be pointed out that the initial orbits used in generating Fig. A-1 are not used elsewhere in this Report. The higher initial orbit used in this study results in effective velocity increments about 1% lower than those shown in Fig. A-1.

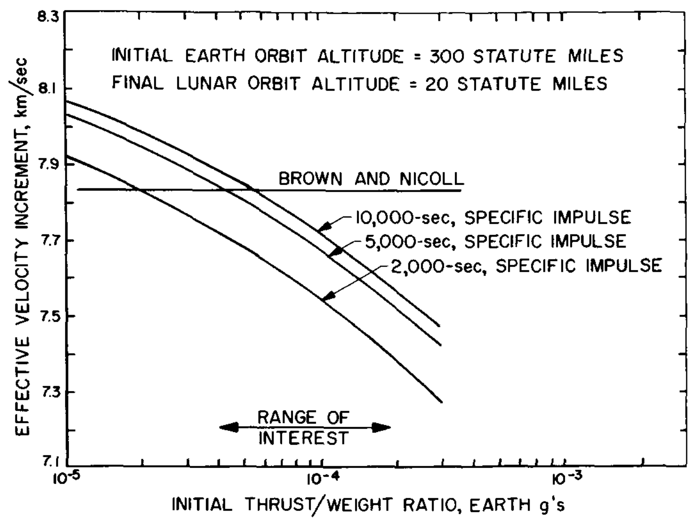

Fig. A-1. Effective velocity increment

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